TECHNICAL NOTES

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

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COMPARISON OF PROFILE-DRAG AND BOUNDARY-LAYER MEASUREMENTS OBTAINED IN FLIGHT AND IN THE FULL-SCALE WIND TUNNEL

By Harry J. Goett and Joseph Bicknell
Langley Memorial Aeronautical Laboratory

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SUMMARY

The effect of the existing turbulence in the full-scale tunnel was determined from measurements of the profile drag of an N-22 section by the momentum method under corresponding conditions in flight and in the tunnel. The transition-point location on the upper surface of the airfoil was also determined from velocity surveys in the boundary layer. The measurements were made at section lift coefficients from 0.480 to 0.635 with a range of Reynolds Numbers from 4,600,000 to 3,900,000.

The results show that the end of transition occurs at approximately the same point on the airfoil in flight and in the tunnel. The transition region was somewhat broader in the tunnel and started farther forward than in flight. The laminar profiles in the tunnel had some characteristics of transition profiles and had a much steeper slope, $\partial u/\partial y$, near the surface than did the laminar profiles obtained in flight. These differences, however, caused an increase of only 0.0001 in the profile-drag coefficients, as determined by the momentum method.

INTRODUCTION

The fact that the profile drag of an airfoil is increased by turbulence in the oncoming stream is well known. This effect is produced by the hastening of the transition from laminar to turbulent flow in the boundary layer, so that a greater portion of the airfoil surface is exposed to the higher turbulent skin friction. Present knowledge is not sufficient to predict quantitatively the drag increment caused by a given amount of turbulence. This investigation was therefore instituted to determine the effect of the existing turbulence in the N.A.C.A. full-scale tunnel (turbulence factor, 1.1) on a conventional airfoil.

The tests were conducted on a wing section of approximately N-22 profile on a Fairchild 22 airplane under corresponding conditions in flight and in the full-scale tunnel. The transition point on the upper surface of the wing was located by means of boundary-layer measurements and the profile drag was measured by the momentum method.

SYMBOLS

The following symbols are used throughout the report:

u, local velocity in boundary layer.

U, velocity at edge of boundary layer.

U_o, free-stream velocity.

s, distance along airfoil surface from forward stagnation point.

c, wing chord.

y, distance above airfoil surface.

c_l, section lift coefficient.

c_{d_o}, section profile-drag coefficient.

TESTS

Apparatus.- The N.A.C.A. full-scale wind tunnel is described in reference 1 and the determination of its turbulence factor of 1.1 by sphere tests is given in reference 2.

The airplane on which the tests were made was a Fairchild 22 parasol monoplane. A panel extending 4 feet along the span of the left wing was covered with 1/16-inch-thick aluminum sheets fastened to heavy wooden ribs for the tests. (See figs. 1 and 2.) The airfoil section at this panel was approximately the N-22 profile; the measured ordinates are given in table I. The chord of the panel was 67.25 inches. Particular attention was paid to obtaining a smooth fair surface around the leading edge

and on the upper surface. Throughout the tests the surface was polished to a high gloss. The lower surface was possibly subject to some small interference from the airplane lift and jury struts, as well as from slight variations in surface finish arising from a removable cover plate.

The boundary-layer measurements were made with two racks, each having four impact tubes and a static tube. The impact tubes were made of hypodermic tubing having an outside diameter of 0.040-inch and a wall thickness of 0.003 inch; they were flattened until their outside depth was 0.012 inch. A photograph of the racks installed on the wing is shown in figure 2. The racks were set approximately 5 inches to each side of the test-panel center line, so that the rear rack was out of the wake of the forward rack. The pressure distribution was determined by static orifices in the wing 10 inches inboard of the test-panel center line and by the static tubes on the racks. The pressures were photographically recorded in flight; in the tunnel, they were read directly on a micromanometer.

In flight, the stream impact and static pressures were obtained from two pitot tubes and a static tube mounted on a boom, one chord length ahead of the leading edge of the right wing, these tubes having been calibrated against a suspended air-speed head. The same reference pressures were used in the tunnel with a small correction for the difference in pressures between the location of the boom and the test panel.

Profile-drag coefficients were measured by the momentum method. The pitot tube and the static tube used to survey the wake were mounted on an arm that swung through an arc about an axis nearly parallel to the chord line of the wing. (See fig. 1.) The traversing arm was mechanically operated from the pilot's cockpit. A locking mechanism stopped the arm at definite increments; its position was recorded. Readings were taken at about 25 stations in the wake (approximately 0.2 inch apart) and at 10 stations outside the wake. The plane of the wake surveys was 5.50 percent of the chord behind the trailing edge. The pressures were recorded in flight and read directly on a micromanometer in the tunnel.

Method.- Boundary-layer surveys at stations ranging from 20 percent to 45 percent of the chord were made at indicated air speeds of 82, 86, and 91 miles per hour in

level flight. Pressure distributions were obtained over the forward 50 percent of the upper surface in order to identify the section attitude and establish the U/U_0 distribution. An attempt was then made to reproduce these flight attitudes in the tunnel by matching the pressure distributions. Additional pressures were measured in the wind tunnel at each attitude to determine the section lift coefficient. The boundary-layer surveys were then repeated in the tunnel at attitudes corresponding approximately to those of flight and at two test speeds that bracketed those of flight.

At an indicated air speed of 86 miles per hour in level flight, momentum surveys were obtained in the wake of the wing with a 0.010-inch-diameter thread on the upper and the lower surfaces at 5 percent of the chord, with the thread on the upper surface removed, and with both threads removed. These measurements were repeated in the tunnel at three test speeds in the range of those of flight.

A summary of the test conditions and results in flight and in the tunnel is given in table II.

RESULTS AND DISCUSSION

A comparison of the chordwise U/U_0 distributions obtained in flight and in the tunnel is shown in figure 3. The approximate section lift coefficient corresponding to each attitude was obtained from pressure distributions over the upper and the lower surfaces of the airfoil measured in the tunnel. It will be noted that exact correspondence in attitude was obtained at values of section c_l of 0.530 and 0.580. The flight attitude giving a c_l of 0.635 was not reproduced nor was the tunnel attitude giving a c_l of 0.480.

The method of determining the transition point by observations of the velocity throughout the boundary-layer profile at several distances from the surface, as discussed by Jones in reference 3, has been used. Plots of velocity distributions along the surface of the airfoil are shown in figures 4 to 7. Since very little variation was noted in the profiles at the two test speeds in the tunnel, the velocity distribution corresponding to the Reynolds Number closer to that of the flight tests is

given. It will be noted that, under corresponding conditions, the end of transition occurs at approximately the same point both in flight and in the tunnel. The transition region, however, is somewhat broader in the tunnel than in flight. With a decrease of c_l, the transition in the tunnel becomes sharper and shows a tendency to resemble more closely that obtained in flight.

The most significant difference between tunnel and flight conditions seems to lie, not in the location of the transition, but in the boundary-layer profiles forward of this point; these profiles, obtained from a cross plot of figure 6, are shown in figures 8 and 9. It will be noted that the velocity at the surface tube (effective center, 0.008 inch from the surface) is consistently lower in flight than in the tunnel. In the boundary-layer profile obtained in flight, this lower velocity shows up as a reflex curvature near the surface, which is characteristic of a laminar profile in a positive pressure gradient, the point of laminar separation occurring where the slope $\partial u/\partial y$ becomes zero or negative in the region close to the surface. In contrast, the boundary-layer profiles obtained in the tunnel have some characteristics of transition profiles for a considerable distance before the transition point. These profiles do not show an inflection point near the surface nor do they have the very high slope at the surface that is characteristic of a fully developed turbulent velocity distribution.

This difference in profiles results in a 12-percent increase in the total skin friction up to the 0.30c station (a Δc_{d_0} of less than 0.0001) as determined from integrations of the momentum loss in the boundary layers at this point. This same difference (0.0001) exists after transition at the 0.45c station and is much less than would be expected to result from the comparatively high $\partial u/\partial y$ at the surface of the airfoil in the tunnel. This discrepancy may be due to the possible existence of velocity fluctuations in the boundary layer, resulting in a high determination of average velocity by the impact tube and a consequent exaggeration of the $\partial u/\partial y$ differences.

From the velocity distribution given in figure 3, the laminar boundary-layer separation point was estimated by a simplified method of applying the von Kármán-Millikan computations (references 4 and 5). The separation points are noted by arrows in figure 3; the transition points for flight and tunnel are also given for comparison. It is

significant to note that, in flight, transition occurs close to separation for the highest c_l tested and moves progressively forward of the separation point as the Reynolds Number is increased (c_l decreased). The tunnel tests show transition occurring at approximately the same distance forward of the computed separation point for all lift coefficients and corresponding Reynolds Numbers.

The profile-drag measurements obtained in flight and in the tunnel at an approximate c_l of 0.580 are shown in figure 10. The method of Jones (reference 6) was used to compute the drag from the impact- and the static-pressure surveys in the wake. Since certain corrections that must be applied because of tunnel effects are not necessary in flight, it was deemed advisable to check the correspondence of the drag measurements under conditions known to be similar. This check was made by a comparison of the measured drags of the section with a thread on the upper and the lower surfaces at 5 percent of the chord; similar flow conditions were thus assured by fixing the transition point on both surfaces. The 0.0001 difference in measured c_{d_0}, which will be noted in figure 10, indicates the difference that may be attributed to the testing technique. With the thread on only the lower surface, a difference in drag of 0.0001 was obtained; and, with both threads removed, the difference was 0.0002. Thus, for this airfoil at a section lift coefficient of 0.580, the change in profile-drag coefficient that may be attributed to tunnel turbulence is 0.0001, which is within the experimental accuracy of the testing technique.

The study of the effect of turbulence was confined to the upper surface because its effect on the flow over the lower surface was expected to be small. This assumption is confirmed by the small change in c_{d_0} produced by fixing transition on the lower surface at 5 percent of the chord (by means of the thread) as well as by the fact that the computed laminar separation point falls between 5 percent and 10 percent of the chord.

The generalization of the results obtained in this investigation must await further information regarding the variation of the effects of turbulence with such variables as pressure distribution, Reynolds Number, and lift coefficient. Such sections as the Clark Y and the N.A.C.A. 4412, within certain limited ranges of the lift coefficient, have a sufficient similarity of pressure distribu-

tion to permit application of the results. Insufficient data are at hand to predict to what extent the results would be modified by such differences in pressure distribution as exist on the N.A.C.A. symmetrical series or on the N.A.C.A. 230 series.

Langley Memorial Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Langley Field, Va., February 2, 1939.

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2. Platt, Robert C.: Turbulence Factors of N.A.C.A. Wind Tunnels as Determined by Sphere Tests. T.R. No. 558, N.A.C.A, 1936.

3. Jones, B. Melvill: Flight Experiments on the Boundary Layer. Jour. Aero. Sci., vol. 5, no. 3, Jan. 1938, pp. 81-94.

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5. Von Doenhoff, Albert E.: A Method of Rapidly Estimating the Position of the Laminar Separation Point. T.N. No. 671, N.A.C.A., 1938.

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TABLE I

Ordinates of Test Panel

(All values in percent of chord)

Station	Upper	Lower
0	4.70	4.70
1.25	5.68	1.62
2.5	6.60	1.12
5	8.21	.48
7.5	9.33	.19
10	10.17	.09
15	11.30	.02
20	12.00	0
25	12.36	0
30	12.40	.03
40	12.02	.12
50	11.07	.31
60	9.56	.40
70	7.69	.43
80	5.49	.36
90	2.92	.21
95	1.56	.10
100	(.15)	0
100	-	0

TABLE II

Summary of Tests

Indicated air speed (m.p.h.)	c_l at test section	Reynolds Number (millions)	Beginning of transition (percent s/c)	End of transition (percent s/c)	c_{d_0} Thread of 0.010-in. diameter at 0.05c, upper and lower surfaces	c_{d_0} Thread of 0.010-in. diameter at 0.05c, lower surface only	c_{d_0} Plain wing
Flight results							
82.4	0.635	4.1	35.5	41.0	-	-	-
86.0	.580	4.3	36.0	42.0	0.0094	0.0074	0.0071
91.2	.530	4.6	37.0	42.0	-	-	-
Tunnel results							
80.6 and 84.7	0.580	3.9 and 4.1	31.0	40.0	[a]0.0095	[a]0.0075	[a]0.0073
84.7 and 88.6	.530	4.1 and 4.3	35.0	41.0	-	-	-
88.6 and 91.6	.480	4.3 and 4.4	39.0	41.0	-	-	-

[a]From figure 10 at Reynolds Number corresponding to flight.

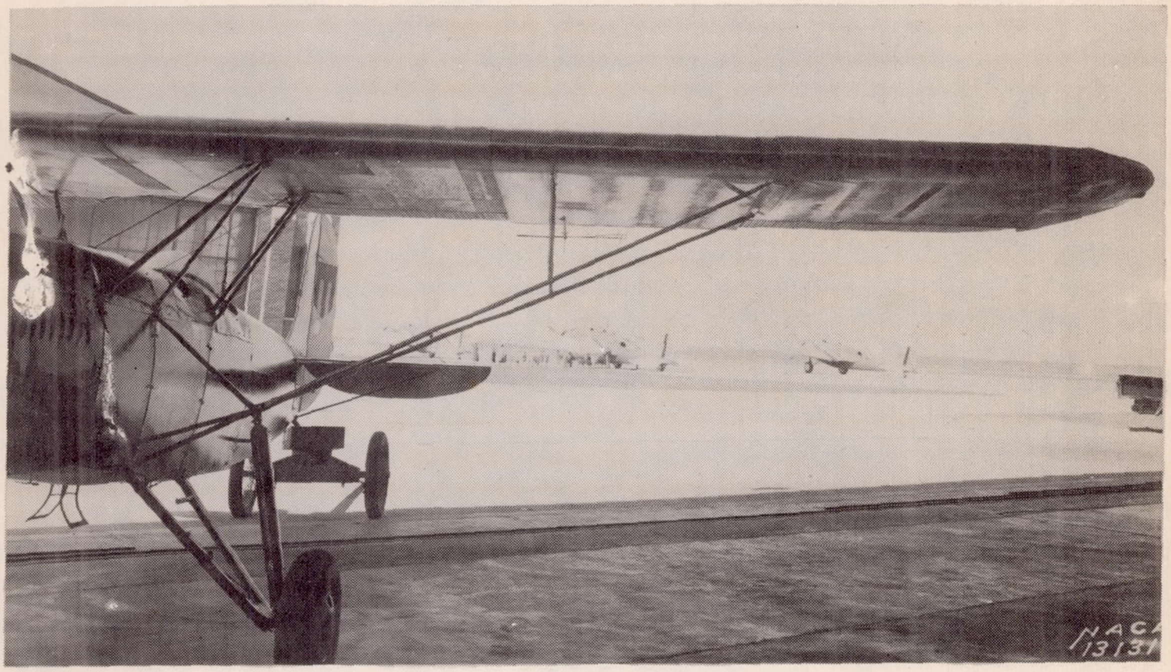

Figure 1.- Test panel on Fairchild 22 airplane.

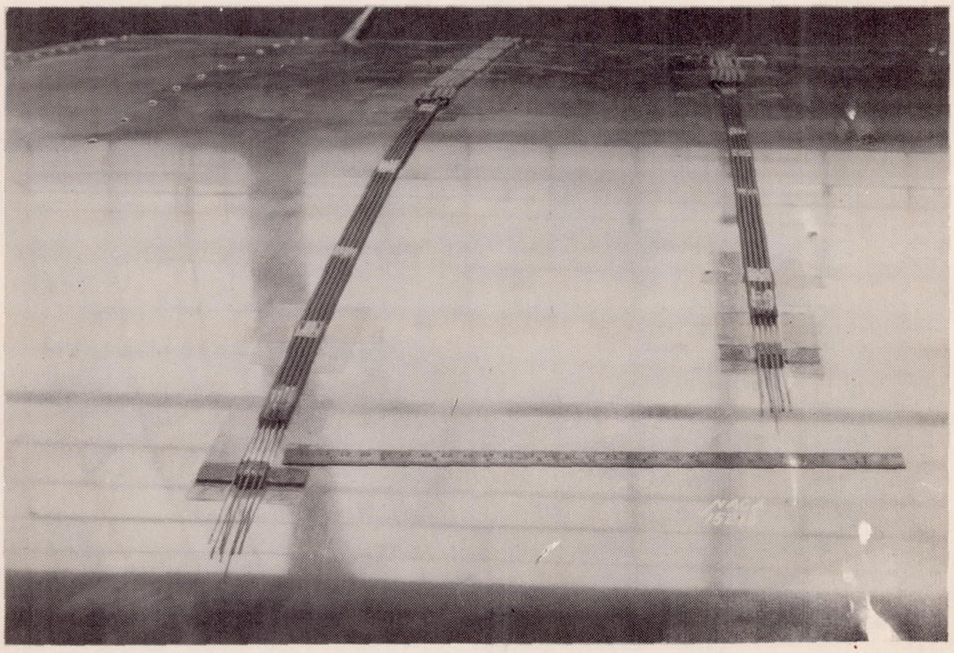

Figure 2.- Boundary-layer survey racks installed on test panel.

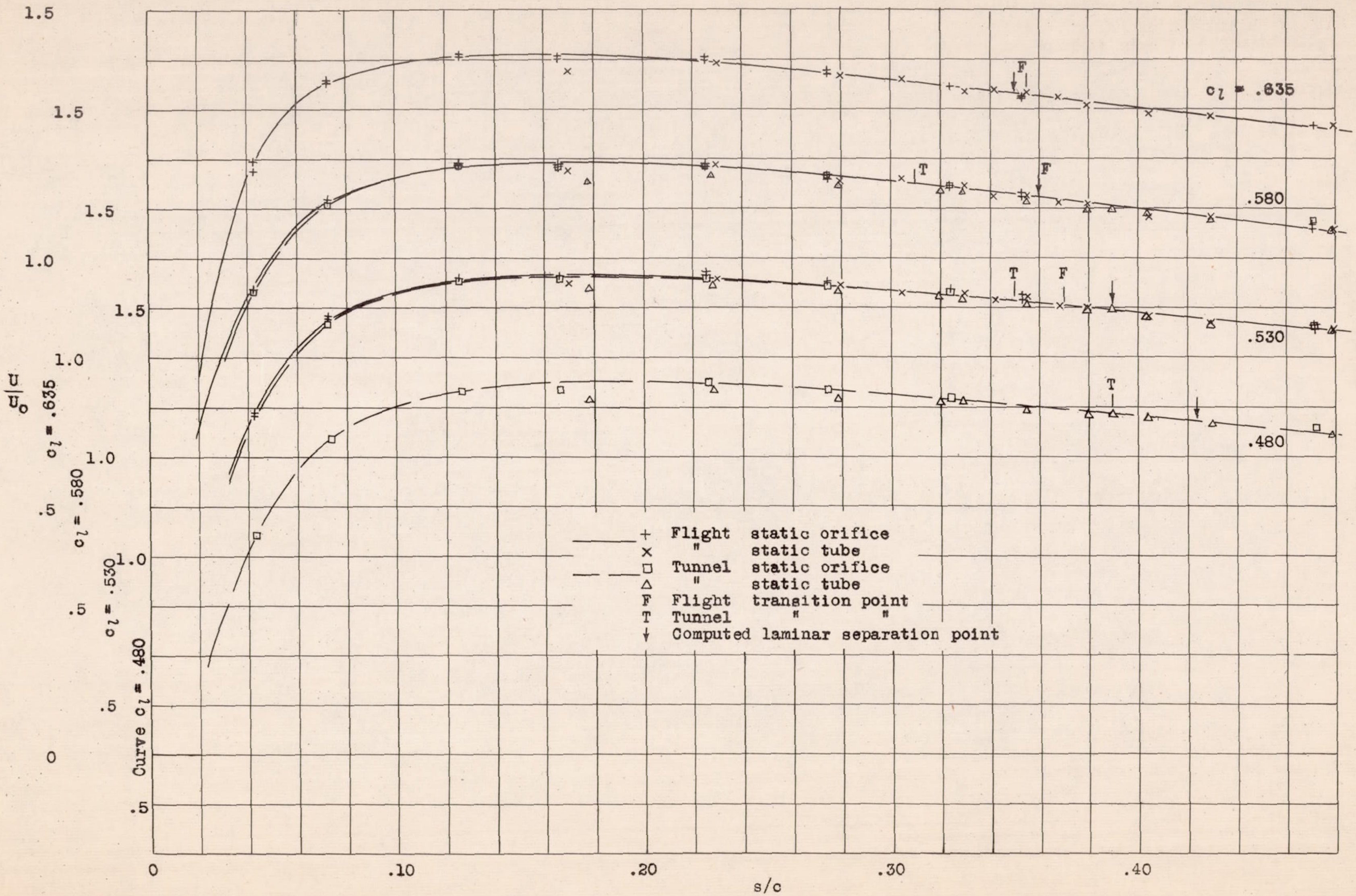

Figure 3.- Velocity distributions along the surface in flight and in the tunnel.

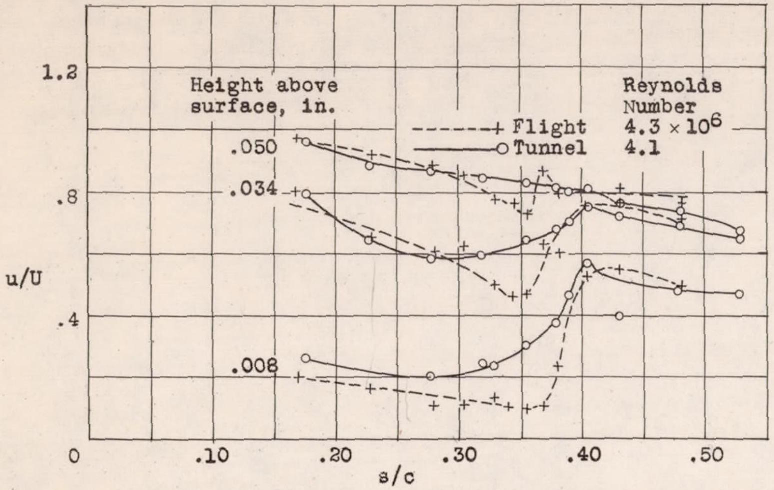

Figure 5.- Boundary-layer velocity distributions along upper surface of N-22 airfoil; c_l , 0.580.

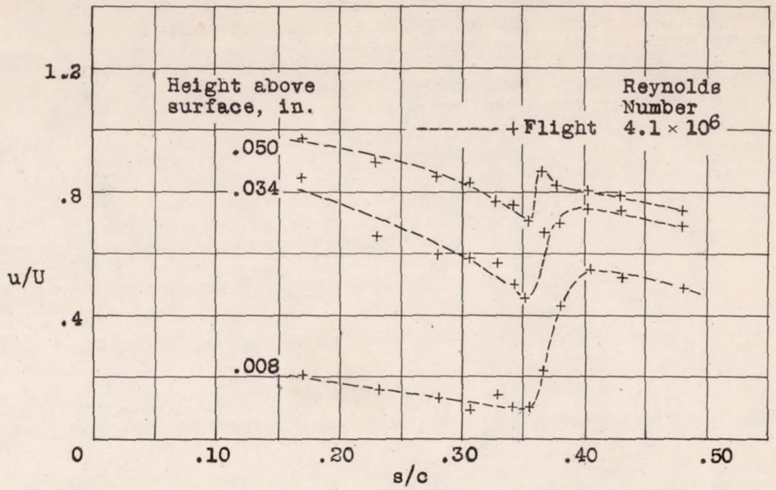

Figure 4.- Boundary-layer velocity distributions along upper surface of N-22 airfoil; c_l , 0.635.

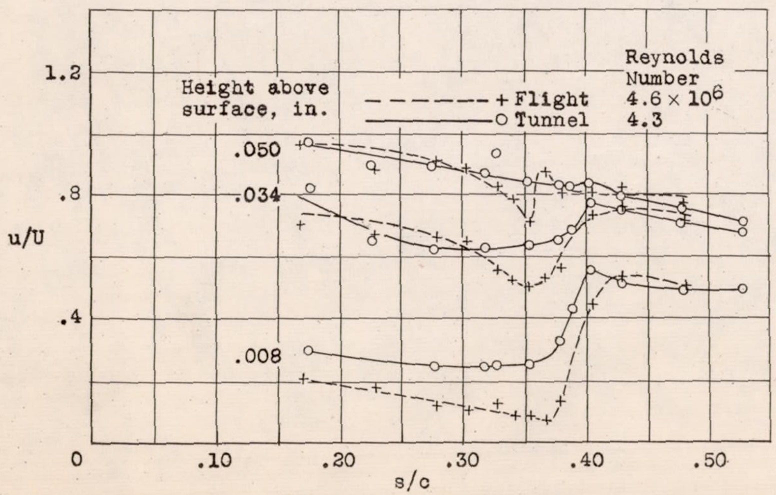

Figure 6.- Boundary-layer velocity distributions along upper surface of N-22 airfoil, c_l , 0.530.

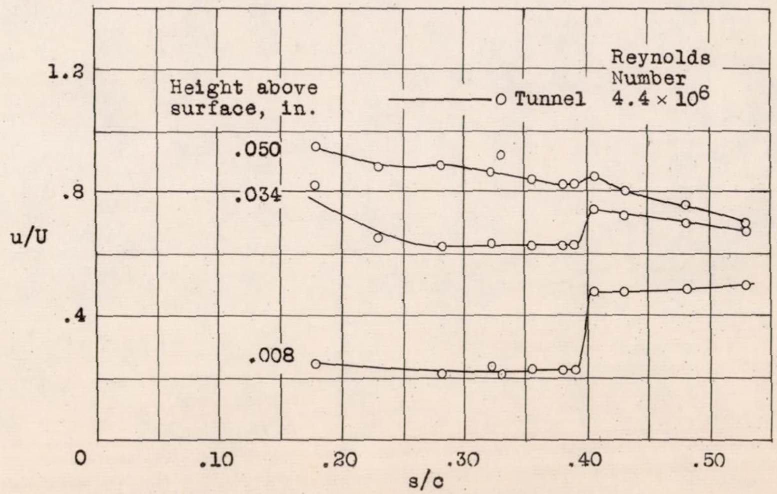

Figure 7.- Boundary-layer velocity distributions along upper surface of N-22 airfoil, c_l, 0.480.

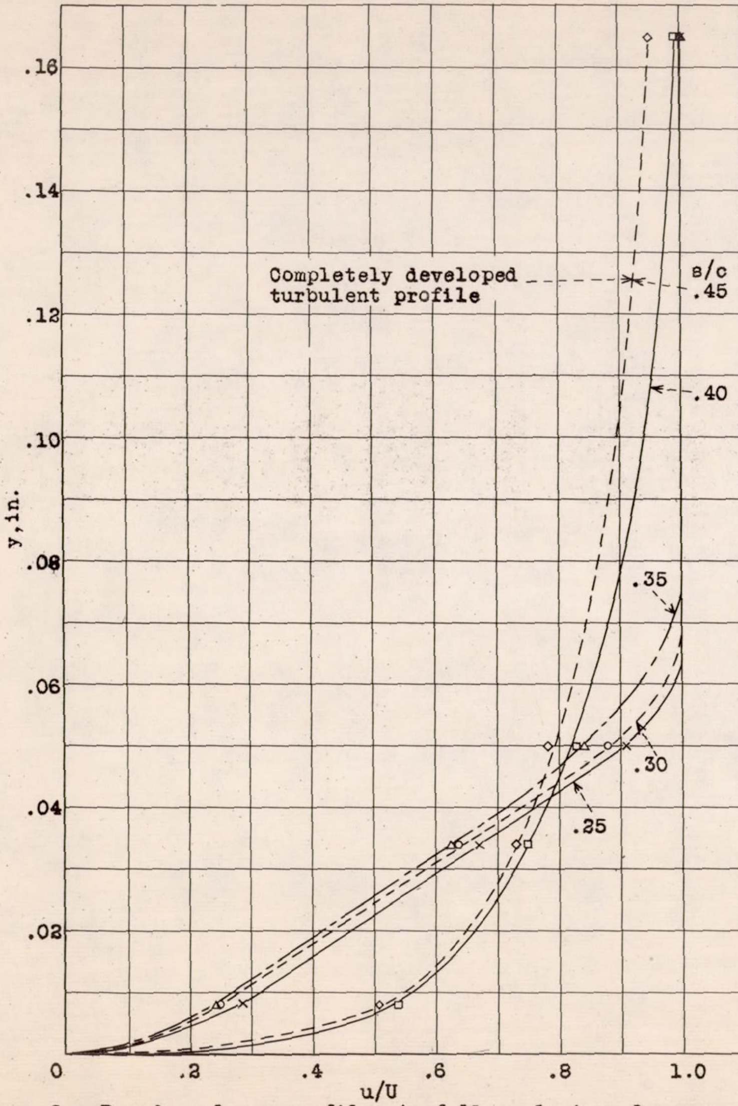

Figure 9.- Boundary-layer profiles in full-scale tunnel on upper surface of N-22 section; c_l , 0.530; Reynolds Number, 4.3×10^6.

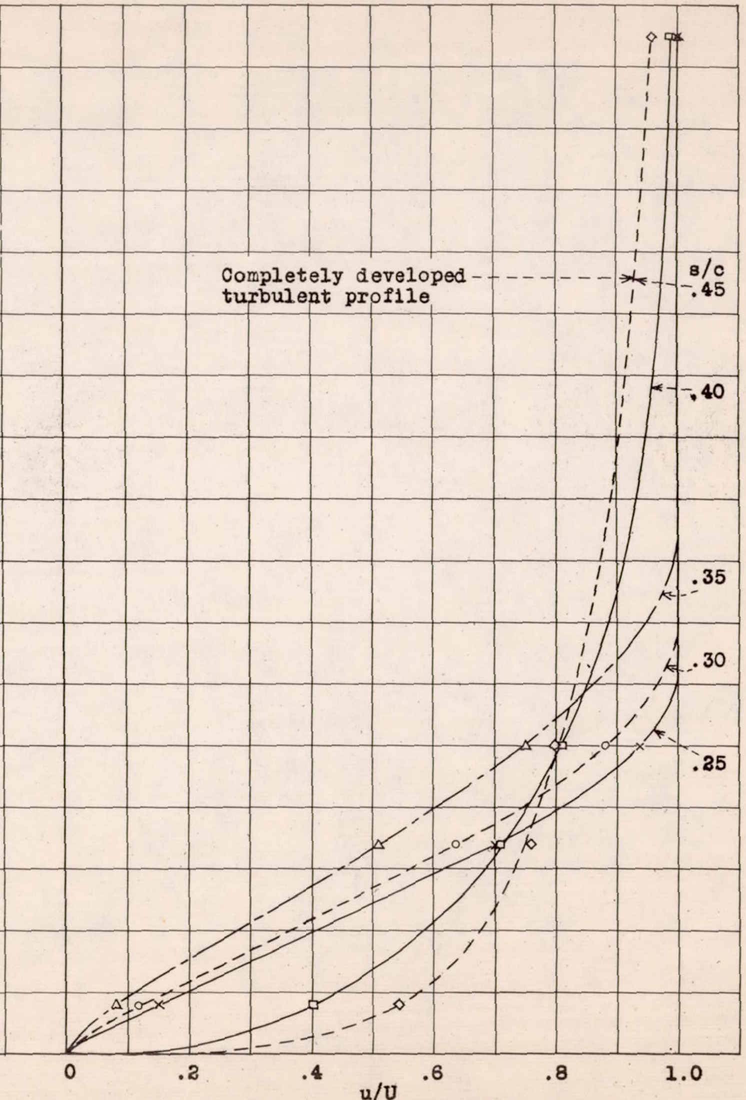

Figure 8.- Boundary-layer profiles in flight on upper surface of N-22 section; c_l , 0.530; Reynolds Number, 4.3×10^6.

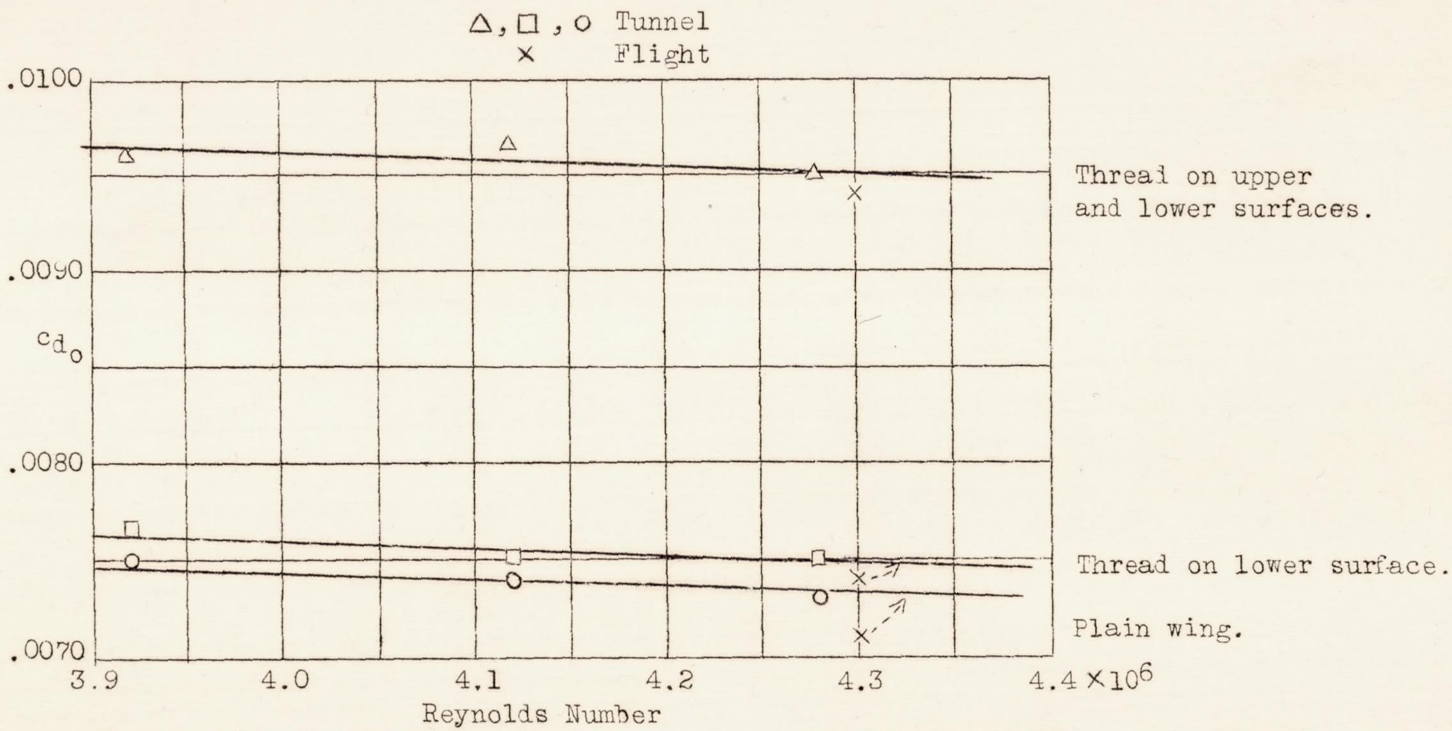

Figure 10.- Results of profile-drag measurements behind N-22 section in flight and in the tunnel.